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by

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# DIMENSION-OPTIMIZING DESIGN METHOD FOR ANNULAR-TYPE COOLING CHANNEL OF THRUST CHAMBER

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ABSTRACT

The new-generation liquid oxygen/hydrocarbon propellant liquid fuel rocket engine will use a high-pressure combustion chamber arrangement. In this case, cooling the thrust chamber becomes a key technical problem. The article presents a design scheme for the geometric-dimension optimization of annular-type regenerative cooling channels. The aim of the optimization is minimum pressure losses as coolant passes through the cooling channel. As shown in typical computations and experiments, application of this optimizing design method can reduce 50% of pressure losses. In other words, the optimization design is advantageous in solving the cooling problem in high-pressure thrust chambers

Key Words: liquid propellant, rocket engine, heat transfer computation, and optimization of cooling channel.

## I. Foreword

In the new-generation liquid oxygen/hydrocarbon propellant

engine, high-pressure combustion chambers are adopted to raise the specific thrust performance of the thrust chamber, as well as to reduce the thrust chamber profile dimensions and mass. As shown in experimental studies, with respect to heat transfer from the combustion gas to the combustion chamber walls, the convective heat flow intensity $q_c$ and the combustion gas mass flowrate (rho·v) or combustion chamber pressure $p_c$ have a relationship of 0.8 times the power index. Hence, an increase in combustion chamber pressure imposes a higher requirement in organizing the cooling process in the thrust chamber. Whether or not the thrust chamber can be reliably cooled becomes a key technical constraint on whether or not the arrangement of the liquid oxygen/hydrocarbon propellant engine with high thrust-chamber-pressure cam be realized. When the heat flow increases during transfer to the chamber wall, in principle the increased amount of heat can be carried away by raising the coolant flow rate, or by reducing the geometric dimensions of the cooling channel, thus raising the coolant flow rate. However, for an engine with a particular thrust, one is limited in utilizing the coolant flow rate approach. Making the coolant flow rate higher will certainly bring about higher pressure losses in the cooling channel, thus adding to the burden of the turbopump system, and affecting the solution of the dynamic equilibrium problem of the entire engine system. On the other hand, making the cooling chamber smaller is constrained by the feasibility of machining the cooling channel. Hence, upon considering the constraining conditions of the cooling requirements and the machining dimensions, a rational design with respect to the cooling chamber dimensions and making adequate use of the cooling potential of the propellant are advantageous in realizing the engine design arrangement for the high-pressure chamber in order to cool the thrust chamber with less pressure drop because of cooling.

In most cases of previous engine development in China, corrugated plate regenerative cooling channels were employed. In

the new-generation liquid oxygen/hydrocarbon propellant engines, annular-type cooling channels will be adopted. In the article, an optimizing design scheme is presented for the geometrical dimensions of annular-type regenerative cooling channels. The purpose of optimization design is to have minimum pressure loss of coolant when passing through the cooling channel.

## II. Simplified Heat-Transfer Model of Thrust Chamber

### 2.1. Convective and radiative heat transfer between combustion gas and chamber wall, and heat conduction through the chamber wall

The convection heat flow intensity $q_c$ can be expressed as

$$q_c = h_c \cdot (T_{aw} - T_{wg}) \tag{1}$$

$T_{wg}$ is the temperature [K] at the chamber wall on the combustion gas side; $T_{aw}$ is the adiabatic temperature [K] of the combustion gas; $h_c$ is the convective heat transfer coefficient, which is determined by the Bartz formula.

In the high-pressure liquid oxygen/hydrocarbon propellant combustion chamber, the combustion gas temperature is as high as 3000 to 4000K, and the chamber pressure is as high as 15 to 20MPa. Here the highly polarized molecules (in the equilibrium combustion products) such as $H_2O$ and $CO_2$ radiate intense radiative heat flow toward the chamber wall. The radiative heat flow intensity can be approximately determined by the method described in [1].

For a heat-resistant stainless steel material with low thermal conductivity, a one-dimensional flat plate assumption can be used to derive the heat conductive flow passing through the chamber wall. However, for an annular-type cooling channel composed of a copper alloy material with a high heat-conducting system, there is an apparent "rib plate" effect that increases

heat conduction on the channel-side wall. Here, the one-dimensional heat conduction model (in [2]) is used to consider the one-dimensional heat conduction model with the side-wall rib effect.

### 2.2 Convective heat transfer between chamber wall and coolant, as well as temperature variation of the coolant

Transmitted to the cooling fluid, the heat flow can be calculated according to the following equation:

$$q_l = h_l \cdot (T_{wl} - T_l) \tag{2}$$

For factors affecting the coefficient $h_l$ for convective heat transfer by the coolant, with the exception of the fluid state and physical properties, effects are present that depend on whether or not the fluid is in a subcritical state or in the supercritical temperature state, and that depend on whether or not a clinkering state exists. For different propellant constituents and different working parameter ranges, many published experiment criterial relationship equations [3, 4] can be applied to determine the $h_l$ value of a propellant.

The entire thrust chamber is divided into N sections along the axial line. Then, from temperature $T_{l,in}$ at the coolant inlet, the temperature $T_{l,out}$ at the coolant exit can be derived:

$$T_{l,out} = T_{l,in} + \sum_{i=1}^{N} [(q_i \pi d_i \triangle x_i)/(\cos\theta_i \dot{m}_l \cdot c_{pl,i})] \tag{3}$$

In the equation, the average values in the section are used for heat flow $q_i$, thrust chamber diameter d, and the included angle $theta_i$ between the wall surface contour line and the axial line of the thrust chamber. The specific heat $c_p$ is obtained by using $T_{l,i}$ as the qualitative coolant temperature at the inlet into the section. $m_i$ is the mass coolant flow.

## 2.3 Pressure drop at the cooling channel

For an incompressible coolant, the following pressure drop formula can be obtained between two random cross sections 1 and 2 of the cooling channel:

$$\Delta P = (\rho_1 v_1^2/2) \cdot [(A_1/A_2)^2 - 1] + f \cdot \Delta x \cdot \rho_1 v_1^2/(D \cdot 2\cos\theta) \qquad (4)$$

In the equation, f is the Fanning coefficient; D is the hydraulic diameter of the cooling channel; delta x is the axial-direction length between cross sections 1 and 2; theta is the included angle between contour and axial line of the thrust chamber.

# III. Dimension Optimization of Annular-type Cooling Channel

## 3.1 Description of the channel-dimension optimization problem

The design target is selecting the minimum pressure drop due to cooling. For a working coolant at a supercritical pressure, satisfying the cooling requirement is a situation when the combustion gas wall temperature $T_{wg}$ is lower than a certain allowable value $T_{wg,max}$, and the coolant wall temperature $T_{wl}$ is lower than the thermal decomposition (or clinkering) temperature $T_{wl,max}$ for the coolant. Fig. 1 is the schematic diagram of the transverse cross-sectional dimensions of the thrust chamber cooling channel. There are the following constraints on machining the cooling channel: wall thickness $t \geqslant t_{min} = 0.65$mm; rib width L $\geqslant L_{min} = 1.00$mm; groove width W $\geqslant W_{min} = 1.00$mm; groove depth $H \geqslant H_{min} = 3.00$mm; depth to width ratio of groove $H/W \leqslant HW_{max} = 4.0$. Then the problem of optimizing the geometrical dimensions of an annular-type regenerative cooling channel can be described as follows:

$$\text{s.t.} \quad T_{wl} \leqslant T_{wl,max} \ , \quad T_{wg} \leqslant T_{wg,max}$$
$$t \geqslant t_{min} \ , \quad L \geqslant L_{min}$$
$$W \geqslant W_{min} \ , \quad 4W \geqslant H \geqslant H_{min} \qquad (5)$$

In the equation, x is the axial-direction coordinate of the thrust chamber; f is the equation expressing the arbitrary function. The cooling-pressure drop delta $p_{cool}$ is obtained from cooling computations.

### 3.2 Dimension-varying shapes of two channels

Beginning from considerations of simplicity in machining the cooling channel, generally it is specified that the basic distribution form of the channel cross-sectional dimensions t, L, W and H is a constant-value form, or a form that linearly varies along the axial line of thrust chamber. Thus, the problem of optimally selecting the four distribution functions, t(x), L(x), W(x) and H(x), is transformed into a problem of optimally selecting several key geometric parameters. The article chooses two-dimensional variation forms for analysis, thus accounting for the importance of reducing the cooling pressure drop by optimizing the design method and the channel dimensions.

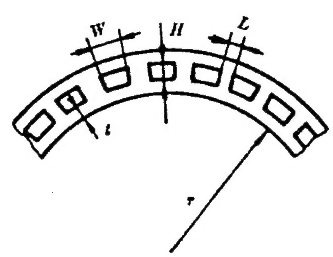


Fig. 1 Transverse cross-section of annular-type cooling channel

(1) The first form (regular design form): let us assume that the distributions of t, L and H along the axial line of the thrust chamber are constants. By adjusting the groove height H

and the groove width $W_1$ at the throat, the minimum pressure drop from cooling is applied to finally determine the design parameters.

(2) The second form (optimized design form): the distribution of t along the thrust chamber axial line is a constant. Refer to Fig. 2 for the variation rule of L and W along the axial line of thrust chamber radius. By satisfying the cooling requirements and machining constraints, the maximum H value should be chosen on each cooling cross section in order to increase the cross-sectional area of the cooling channel and to reduce the cooling pressure drop. After selecting the rib width $L_b$ of the cylindrical section of the combustion chamber and the groove width $W_1$ at the throat, selection of groove height H can be determined by induction with the following optimization problem of a single variable.

$$\max: H(x), \quad \text{s.t.} \quad 4W \geqslant H \geqslant H_{min}, \quad T_{wg} \leqslant T_{wg,max}, \quad T_{wl} \leqslant T_{wl,max} \tag{6}$$

By manipulating $L_b$ and W1, the minimum pressure drops corresponding to $L_b$, W1 and H(x) can be obtained. From Fig. 2, by computations we can obtain L(x) and W(x).

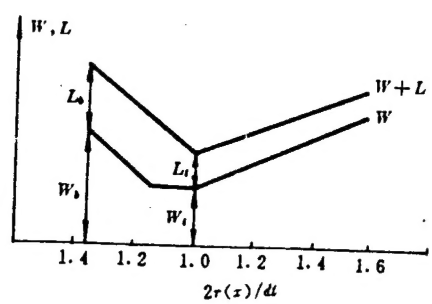


Fig. 2 Distribution of W and L with respect to the inner diameter of the thrust chamber

Here, a set of $LO_2$/RP-1 propellant engine parameters constitutes the initial input parameters in an example of cooling

computations. The combustion chamber pressure $p_c = 16.67$ MPa; the coolant flow $\dot{m}_l = 100.0$ kg/s; the throat diameter dt $= 0.2327$m ; at nozzle exit, the area ratio $epsilon_e$ = 30.0; and the mean excess oxygen coefficient (of thrust chamber) is 0.818. In addition, assume that the excess oxygen coefficient (at the thrust chamber side) is 0.47. Let us assume that the coolant enters the cooling channel at a nozzle area ratio of 8:1; the internal wall thickness is 0.001m; and the material at internal wall is a Zr-Cu alloy. From the derived $d_1$ and $epsilon_e$, the axial-direction variation rule r(x) of the inner diameter of dual circular arc nozzle can be determined by selecting the following: combustion chamber characteristic length $L^o$, combustion chamber convergence ratio $epsilon_c$, circular arc radius $R_1$ at the convergent section inlet of the nozzle, convergence angle $beta_1$, curvature radii $R_{t1}$ and $R_{t2}$ at throat, and the nozzle exit angle alpha.

### 3.3 Comparison of calculated results

(1) Results of the first dimension-distribution form: by taking 0.001m as the rib width, Fig. 3 shows the variation relationship of the cooling-pressure drop with throat groove width $W_1$ and groove height H. Fig. 3(b) shows the variation relationship of the coolant side wall temperature with $W_1$ and H. Within the parameter range in the computations, the temperature at the combustion-gas-side wall is always lower than the allowable value, therefore the temperature is not shown in the figure. If the clinkering temperature at the coolant side is assumed to be 700K [5], $W_1$ can be obtained for each H in Fig. 3(b). Then, according to the H and $W_1$ thus obtained, a cooling pressure drop can be obtained from Fig. 3(a). From Fig. 3(a) and Fig. 3(b), a set of design parameters with relative satisfaction is: $(H, W_t, \Delta P_{cool}) = (0.005\text{m}, 0.00122\text{m}, 4.30\text{MPa})$

(2) Results of the second dimension distribution form:

according to the channel-dimension variation rule as shown in Fig. 2, after selecting $L_b$ and $W_l$, rib width L and channel width W are determined for each cross section of the thrust chamber. By picking a set of ($L_b$, $W_l$) values for the cooling computations, in these values H(x) is chosen for optimization from Eq. (6). In addition, by taking $(T_{wg,max}, T_{wl,max}) =$ (840,700)K. Refer to Fig. 4(a) for the calculated result thus obtained. From Fig. 4(a), in this case the optimal design parameters are the following: $(L_b, W_l, \Delta P_{cool}) = (0.0021\text{m}, 0.002\text{m}, 2.167\text{MPa})$. Comparing the cooling pressure drop here with the result of the first dimension distribution form, the cooling pressure drop can be reduced by 50 percent. Fig. 4(b) shows the distribution situation of the cooling channel height H along the axial direction of thrust chamber after optimization.

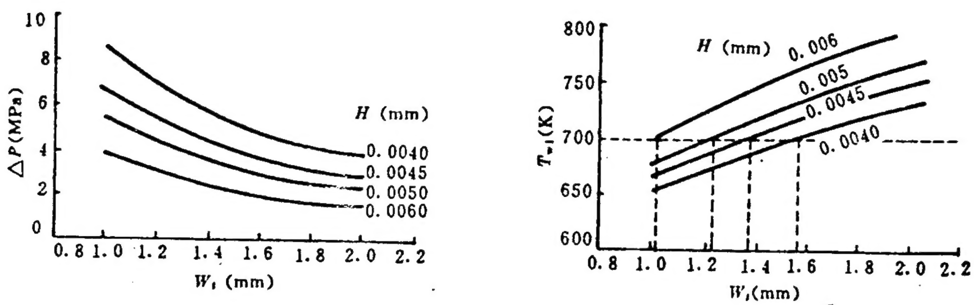


Fig. 3. First type of dimension distribution

### 3.4 Dimension optimization steps for annular-type cooling channel

With induction in the process of calculating the cooling pressure drop to the second dimension-distribution form mentioned above, the following steps in dimensional optimization of the cooling channel are obtained: (1) from the given design parameters of the thrust chamber, determine the relationship r(x), the inner wall radius of thrust chamber as distributed

along the axial direction of thrust chamber. (2) Choose the rib width $L_b$ of the cylindrical section of the thrust chamber, and the groove width $W_1$ at the throat. From Fig. 2 and r(x), L(x) and W(x) are determined. (3) Divide the thrust chamber into several cross sections; from the given heat transfer model, compute the temperatures at the gas wall and at the liquid wall along the transverse cross section. The maximum groove depth is obtained by using Eq. (6). (4) Obtain the summation of cooling pressure drops; thus, the total cooling pressure drop is obtained. (5) Adjust the $L_b$ and $W_1$ values for the further step of cooling computations to find the cooling pressure drop. Plot a computation result diagram as shown in Fig. 4(a). From the figure, the $L_b$ and $W_1$ values corresponding to the minimum cooling pressure drop can be determined. These values are the optimization result. (6) From the $L_b$ and $W_1$ that are obtained and from Fig. 2, the final distributions of L(x) and W(x) can be determined. Since H(x) has been determined by the heat transfer computation process, the L(x), W(x) and H(x) from optimization can be used to guide the actual engineering design of the cooling channel.

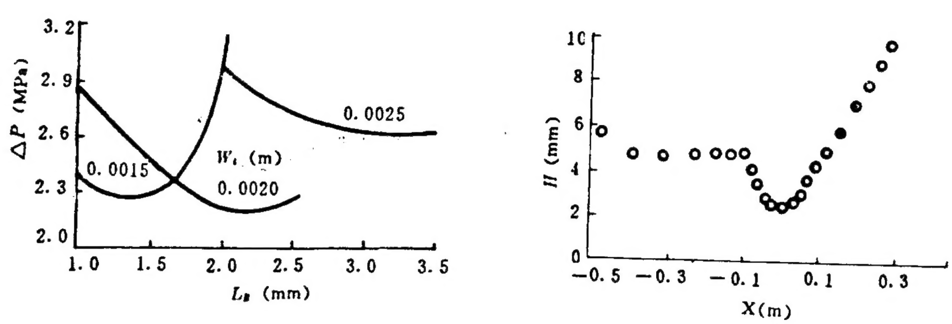


Fig. 4. Second type of dimension distribution

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